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THRUST VARIATION OF A
GASEOUS PROPELLANT
ROCKET ENGINE

THESIS

GAM/ME/67-5

Frederick J. De Groot
1st Lt USAF

SCHOOL OF ENGINEERING

WRIGHT-PATTERSON AIR FORCE BASE, OHIO

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THESIS

Presented to the Faculty of the School of Engineering of
the Air Force Institute of Technology
Air University
in Partial Fulfillment of the
Requirements for the Degree of
Master of Science

by

Frederick J. De Groot, B.S.
1st Lt USAF
Graduate Aerospace - Mechanical Engineering

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Preface

This report represents the third investigation by an AFIT student in the area of rocket thrust variation. A variable thrust gaseous propellant rocket engine incorporating a variable area injector was designed, built, and tested. This engine represents a significant improvement over that of previous designs attempted here at AFIT and should prove to be a stable and reliable tool for future investigations in this field.

In this report, I have assumed the reader has a fundamental background in science as well as an acquaintance with rocket propulsion theory and terminology.

I wish to acknowledge indebtedness to my advisor, Lieutenant Colonel Hamilton, for his timely support and guidance; to Mr. Wolfe and his staff at the AFIT Machine Shop; and to Mr. Parks, the lab technician. Finally, I would like to specially acknowledge and thank my wife, Jeannine, who has contributed much more to this effort than she herself may suspect.

Frederick J. De Groot

Abstract

The purpose of this thesis was to investigate the performance of a variable thrust rocket engine incorporating a variable area injector. The injector was designed, constructed, and assembled on an existing thrust chamber which was lengthened and provided with water cooling. Using gaseous hydrogen and oxygen as propellants, an extensive test program was completed at the AFIT Rocket Engine Test Facility. The thrust was varied over a continuous range from 11 to 75 pounds, resulting in a throttling ratio of 6.82:1, while the specific impulse remained nearly constant. The transient response of the engine was fast, smooth, and accurate, and no indication of combustion instability was observed during the investigation.

Contents

List of Symbols

A	Area, in.^{2}
C*	Characteristic exhaust velocity $P_c A_t g_c / m$, fps
C_f	Thrust coefficient, $F/P_c A_t$
D	Diameter, in.
F	Thrust, lb_f
gc	Conversion factor, 32.2 $lb_m ft/lb_f - sec^2$
I_{sp}	Specific impulse, F/m, lbf sec/lbm
m	Mass flow rate, lbm/sec
MR	Mixture ratio (m_O/m_H)
P	Pressure, psi
R	Radius, in.
T	Temperature, °F, °R

Subscripts

c	Combustion chamber
H	Hydrogen
o	Stagnation
O	Oxygen
t	Nozzle throat

List of Figures

List of Tables

THRUST VARIATION OF A GASEOUS PROPELLANT ROCKET ENGINE

I. Introduction

Background

Today's rapidly advancing space technology continually demands more sophisticated hardware to accomplish its expanding mission requirements. As missions increase in complexity, a high degree of space vehicle control becomes essential. Throttleable rocket engines will be increasingly utilized to achieve this control. The Apollo LEM (Lunar Excursion Module) will make use of such engines to execute its lunar landing maneuvers.

The use of auxiliary systems to perform functions outside the capabilities of constant-thrust, single firing devices represents today's solution to vehicle velocity control. This method could conceivably be used to accomplish the intricate maneuvers associated with lunar and interplanetary missions. However, from the standpoint of over-all system complexity and weight,

use of a single propulsion device with sufficient throttling and restart capability to perform all of the required maneuvers is clearly preferable to the use of a multiple element system.

The basic disadvantage of providing rocket engines with a throttling capability is that such propulsion units must be more complex than constant-thrust rocket engines. Poor propellant mixing, combustion instability, and nozzle separation are also problems which can arise. Therefore, research on the throttling process is necessary for the development of efficient and reliable throttling techniques.

Problem

The purpose of this thesis was to design, construct, and test a rocket engine throttling system. The design objectives were to improve the performance of an existing throttleable rocket engine and to provide an engine suitable for further research on the throttling problem. The test objective was to evaluate the performance and response of the rocket engine with respect to the throttling process. The engine was to be tested at the Air Force Institute of Technology (AFIT) Rocket Engine Test Facility, using gaseous hydrogen and oxygen as propellants. Existing equipment was to be utilized as much as possible.

Previous Work

Very little work has been done on the throttling of gaseous

propellant rockets, liquid propellant rockets having received most of the throttling attention. However, in recent years, there has been some work done at AFIT on rocket thrust variation, using gaseous oxygen and gaseous hydrogen as propellants because of their ready availability.

The first work on thrust variation at AFIT was done by Watkins (Ref 9), who examined four general throttling techniques: a) Throttling valves in the propellant supply lines, b) contamination of the propellants with inert constituents, c) throat area variation, and d) injector area variation. After concluding that injector area variation held the most promise for gaseous propellant rocket engines, he designed a variable area injector for installation on a small film-cooled gaseous rocket engine which had been designed and built by Ow (Ref 4). Watkins also generated a series of theoretical performance curves for this engine, using a computer program written by Anderson (Ref 1).

Following Watkins' work, Smith (Ref 7) evaluated the design, found several disadvantages, and then designed a different throttling system, still using the technique of injector area variation. The injector area for each of the propellants was varied by means of a metal plate which could be moved back and forth over a fixed orifice plate. The movable plates were connected to a shaft which in turn was connected to a throttle arm.

Problem Analysis

After reviewing the previous work done in this field, particularly that of Watkins and Smith, it was decided that an extension of Smith's experimental work was the most logical approach to the problem. Smith's experiments verified that injector area variation is an effective method of controlling the thrust of a gaseous propellant rocket engine. His report included data for nine steady state runs which were conducted at eight different fixed throttle positions. Four transient runs were reported, but there was no instrumentation provided which would indicate the throttle position or the injector area as a function of time. In fact, throttle slippage was observed during some of the runs, thus precluding even an estimate of injector area. Therefore, it was decided to more thoroughly investigate the throttling process as it affects the performance and transient response of a gaseous propellant rocket engine.

Smith's design incorporated several salient features:

1. The orifices, each of which consisted of a flat brass plate, were removable and easily made. Thus, with a set of different orifice plates, one could vary the mixture ratio from run to run.

2. The doors which slide across the orifice plates were located on the inside of the propellant manifolds. Thus, they were held against the orifice plates by the differential pressure

across them and helped maintain an effective seal.

3. The throttle was completely variable to any position from fully closed to fully open.

4. The design was simple and contained only a minimum of moving parts.

There were several shortcomings in Smith's design which could not be corrected by modification of the existing equipment. Therefore, it was decided to design a new throttling mechanism incorporating the desirable features of Smith's design, and using as much existing equipment as possible.

II. Equipment Design and Construction

Design

The overall design of the throttling mechanism was kept as simple as possible. Basically, it consists of an injector incorporating two separate orifices, one for hydrogen and one for oxygen. The areas of the propellant orifices are varied by means of two plates which slide over the orifices. The moveable plates are connected to a single throttle shaft, and therefore, move simultaneously as the shaft is turned by a throttle arm. The details of the injector can be easily seen on the photographs in Figures 1 through 5, and on the drawings contained in Appendix A.

The throttle mechanism is actuated by a hydraulic slave unit which is connected to a command unit in the control room. This system provides positive manual control of the throttling mechanism during engine operation. A schematic diagram of the hydraulic control system and a brief outline of the procedures used to prepare it for operation is contained in Appendix B.

The similarities between this design and Smith's design are readily apparent. However, this design corrects several shortcomings or improves upon several aspects of Smith's design. Since this redesign could not be accomplished by modification of the existing equipment, the design and construction of a new injector was necessary. The desire to utilize as much

existing equipment as possible in the interests of economy and time led to the design of an injector which bolts directly to the existing thrust chamber. The face of the injector is flat with one opening in the center through which the oxygen is injected. Ten holes through which the hydrogen is injected are arranged in a circle around the central hole. The hydrogen is injected at an angle of 30 degrees to the centerline of the chamber and impinges on the oxygen flow approximately two inches from the injector face. The impinging flow injector design was chosen to promote better mixing of the propellants, and thus, to increase the combustion efficiency of the engine. Also, this design completely eliminates the momentum problems encountered during the previous investigation. There is no need to calculate the momentum of the propellant flows at any mixture ratio. The oxygen flow has only axially directed momentum and because of the symmetrical arrangement of the hydrogen injection holes, the radial components of momentum of the hydrogen flow cancel each other for any flow rate or mixture ratio. Thus, the injection geometry tends to keep the center of combustion away from the chamber walls and the injector face, and by so doing, reduces the possibility of hot spots or burnout. This injection scheme necessitated the addition of propellant manifolds downstream of the throttling orifices, thus increasing the weight of the injector. However, considering the use of the system only

as a laboratory research tool, the weight penalty was considered minor in view of the advantages to be gained.

The propellant orifices, each of which consists of a flat brass plate, are removable and easily made. This feature permits experimentation with different mixture ratios and flow rates simply by changing orifices. For this investigation, the orifices were lengthened and significant changes to their geometry were made. The orifice openings of the old design were rectangular in shape with length to width ratios of 14.1:1 and 6.5:1 for the oxygen and hydrogen orifices, respectively. Also, the lengths of the two openings were not equal. This configuration resulted in a varying oxygen to fuel injection area ratio as the throttle plates pivoted over the propellant openings, which caused the mixture ratio to change with throttle position. The new orifice openings were of the same length and were curved to coincide with the arc swung by the throttle plates. This modification resulted in a constant area ratio between the two openings over the entire throttling range. The length to width ratios of this orifice design are 25.6:1 and 12.8:1 for the oxygen and hydrogen orifices, respectively. The thinner orifice openings allowed more precise control of the propellant flow areas.

The propellant manifolds were made larger to allow the throttle plates to clear the propellant orifices before striking

the manifold wall. The throttle shaft was made 1/2 inch in diameter and a .188 inch diameter hole was provided radially at the center of the shaft. A 2 1/2 inch machine screw inserted axially through the center of the throttle arm fit into this hole and eliminated the throttle slipping which occurred in the previous design.

The use of separate manifolds for the hydrogen and oxygen eliminated the danger of propellant crossleakage. Thus, no internal sealing was required in the injector. Two O-rings, held in place by steel pressure plates, provided the external sealing where the throttle shaft entered the propellant manifolds. This shaft seal was the only dynamic seal required in this design.

Two modifications were made to the chamber:

1. A 1 3/8 inch sleeve of the same material as the chamber was inserted between the aft end of the chamber and the nozzle. This resulted in a 20% increase in the characteristic chamber length (L^*). This change was made in the interest of attaining complete combustion of the propellants within the combustion chamber, thus making the assumption of frozen flow through the nozzle more valid. Optimizing L^* was not considered to be important for a laboratory test engine of this type.

2. A water cooling capability was added to the combustion chamber in the form of three separate coils of 1/4 inch copper

tubing which were wrapped around and soldered to the chamber. This tubing is in addition to the tubing previously installed on the nozzle. The use of four smaller coils instead of one large coil results in a greater mass flow rate of coolant without the additional complexity of a pressurization system for the water coolant supply. The building water supply pressure is approximately 80 psig at the test facility. The cooling was provided to help maintain wall temperatures within safe limits and also to reduce the time period required for cool-down between runs.

Construction

The injector components were built and assembled in the AFIT Machine Shop. The basic structure of the injector was made of type 304 stainless steel because of its high temperature properties and resistance to oxidation. The components of the basic structure, the base plate, the manifold jackets, the manifold spacer, and the injector covers were arc welded together. This construction technique insured no crossleakage of propellants.

The combustion chamber extension was made of type 1060 cold rolled steel, the same type as the chamber itself. This type steel has slightly lower strength and melting point characteristics than stainless steel, but it offers the advantage of higher thermal conductivity. This feature increases the ability of the cooling system to limit the chamber wall temperature.

Asbestos gaskets were used as seals between the engine components, which were assembled with high strength aircraft bolts.

III. Thermochemical Data

Theoretical Performance

The engine performance at a mixture ratio of 2.0 with a chamber pressure of 250 psia was chosen as the full throttle base line performance with which to compare the effects on performance of the designed throttling method. This mixture ratio and chamber pressure were picked to be the same as in the previous design so that the same thermochemical data would apply. This mixture ratio was chosen in the previous study to limit the flame temperature (2075°), thus reducing the possibility of burnout. The chamber pressure is well within the safety limits of the equipment and was selected in the interest of propellant economy.

The nozzle has a 0.514 inch throat diameter and an exit to throat area expansion ratio of 4.24:1 with a 15 degree half-angle of divergence. Using a computer program written by Anderson, Watkins generated theoretical performance data for this nozzle at various mixture ratios and chamber pressures (Refs 1 and 9). The basic rocket performance parameters of C^*, C_f, and I_{sp} were calculated for expansion over the given nozzle expansion ratio into an ambient pressure of 14.5 psia (the average ambient pressure at the test facility) rather than for expansion to a specific nozzle exit pressure, which is

the common practice when presenting theoretical performance. With this type of calculation, the theoretical data can be compared directly with the experimental results.

The assumptions used to obtain the theoretical performance data are the following:

1. The combustion process is adiabatic.
2. Combustion occurs at constant chamber pressure.
3. The propellants are introduced into the chamber in gaseous form at a temperature of 298.15°K.
4. Homogeneous mixing of the propellants is attained.
5. Thermal and chemical equilibra exist in the combustion chamber.
6. The products of combustion have zero bulk velocity in the combustion chamber.
7. All species behave as ideal gases.
8. Dalton's Law is applicable.
9. Nozzle flow is steady, isentropic, and one dimensional.
10. The relative concentration of the combustion products remains constant during the nozzle expansion (frozen flow).

These assumptions are generally accepted in the field as appropriate for theoretical calculation of rocket engine performance (Ref. 8).

Experimental Data

All data was collected on a Consolidated Electronics Corporation recorder and reduced on the AFIT IBM 1620 computer, using a simplified version of a program written by Anderson (see Appendix C). The program output data included the following quantities: Run number, throttle position, thrust, mass flow rates, mixture ratio, chamber pressure, characteristic exhaust velocity, thrust coefficient, and specific impulse.

IV. Experimentation

Test Objectives

The purpose of the experimental effort was to determine the performance of the variable thrust rocket engine through its throttling range for the designed mixture ratio of 2.0. These results were to be compared with the engine performance predicted by the theoretical analysis in order to determine the effect of the throttling method on the actual rocket engine performance. The maximum throttling capability of the engine was to be demonstrated. Finally, the transient response of the engine was to be determined.

Apparatus

The experimental testing of the variable thrust engine was conducted at the AFIT Rocket Test Facility, Building 79-D, Wright Patterson Air Force Base. The major components of the facility include:

1. Two propellant manifolds for gaseous hydrogen and gaseous oxygen and a gaseous nitrogen manifold to provide a purge and control system.

2. A control room (Figure 6) with a test console to control and synchronize the test sequence, and a Consolidated Electronics Corporation recorder for recording test data.

3. A propellant control system (Figure 7) with check

valves, dome pressure regulating valves, and solenoid valves for controlling the mass flow rates and pressures.

4. Two Herschel venturi meters with associated pressure transducers and thermocouples for measurement of the data from which the propellant mass flow rates were calculated.

5. Necessary piping, wiring, instrumentation, and calibration equipment.

A detailed description of the above components is presented in the Facility Operations Manual (Ref 3).

The throttle was set manually for each steady state run. The hydraulic control was used only during transient runs at which time the throttle position was measured by means of an electric potentiometer attached directly to the throttle arm.

Procedure

1. Leak Check: The thrust chamber assembly, as originally designed, was checked for leaks after plugging the nozzle and pressurizing the chamber to 75 psig with nitrogen. Leakage was noted at all gasket surfaces when the leak test solution was applied, and leakage past the throttle shaft O-rings was excessive and considered dangerous. The chamber leakage was corrected by sealing the chamber joints with electric heater element cement, but attempts to stop the throttle shaft leakage by using larger and stronger seal plates

were unsuccessful. Therefore, the injector was disassembled and .500 inch inside diameter, .100 inch height rubber O-rings were installed around the throttle shaft inside the propellant manifolds. The O-rings were countersunk into the manifold walls (See Figure 19 in Appendix A), held in place by brass plates, and lubricated with Airco #20 Lubricant which is suitable for oxygen service. The engine was then reassembled and leak tested at chamber pressures in excess of 300 psig with no leakage detected.

2. Cold Flow Runs: The first set of runs consisted of nine (9) full-throttle cold flow runs in which the propellants were flowed through the engine simultaneously but were not ignited. This series of runs was accomplished in order to make an initial determination of the propellant supply line pressures and the pressure regulator loader settings required to obtain the desired propellant mass flow rates.

3. Steady State Hot Flow Runs: A series of approximately twenty (20) full throttle runs were made in which the propellants were ignited in order to determine more exactly the required propellant supply line pressures and the pressure regulator loader settings. Once the proper line pressures were determined, they were held constant throughout the remainder of the experimental testing program.

The first few runs verified that the water cooling was adequate, since full throttle runs of up to 7.5 seconds duration resulted in no damage to the engine. Since only 3.5 seconds were required for the propellant mass flow rates and the engine performance parameters to reach steady values, a standard run time of approximately 5 seconds duration was established.

During this initial set of runs, an occasional "pop" sound was heard over the test cell intercom monitor during the purge sequence. Investigation revealed that the hydrogen propellant line was hot in the vicinity where the purge gas entered the propellant line, indicating that the air which had been substituted for nitrogen because of its ready availability, and hydrogen were igniting in the propellant line. In the interest of safety, the use of air as a purge gas was discontinued, and it was replaced by nitrogen. No further difficulties with the equipment were encountered during the remainder of the investigation.

Following the initial set of runs, eighty-eight (88) steady state runs were made at various throttle settings while attempting to hold the line pressures at 324 psig and 282 psig for the hydrogen and oxygen, respectively.

4. <u>Transient Runs</u>: The hydraulic throttle control was activated and ten (10) runs were made with cycling of the throttle setting during each run while propellant supply pressures

were again held constant. Run times of 8 seconds allowed steady state conditions to be reached at the initial throttle setting, and permitted 2 seconds for throttling in each direction. Some runs were made in which the throttle was moved through its range in only 1 second.

V. Results and Discussion

Steady State Operation

The steady state data are tabulated and presented in Tables I through VIII, one table for each of the eight throttle settings used. Each table includes the results of eleven (11) runs made at that particular throttle setting and the average of the results for those eleven runs. The average thrust, mixture ratio, characteristic exhaust velocity, thrust coefficient, and specific impulse are plotted versus throttle setting in Figures 8 through 12.

The effects of injector area variation as a method of thrust variation on the performance of the gaseous propellant rocket engine used in this study are readily apparent from Figures 8, 9, and 10. The characteristic exhaust velocity rises slightly at the lower end of the throttling range while the thrust coefficient drops slightly. These two effects tend to counter each other, thus causing the overall performance parameter, the specific impulse, to remain nearly constant throughout the entire range of throttling.

The mixture ratio (Figure 11) does not remain constant over the throttling range as was desired for the investigation. The injecotr was designed so that the ratio of the two propellant orifice areas remained constant throughout the throttling range.

However, the relative mass flow rates of the propellants vary somewhat as the pressure drop across the orifices is increased at the lower thrust levels.

The thrust versus throttle position curve (Figure 12) is non-linear and indicates that most of the thrust variation occurs over the lower half of the throttle range. As a demonstration of the throttling capability of the system, two runs were made with the throttle in the closed position. In this configuration, the engine idled smoothly and showed no signs of combustion instability while producing 11 pounds of thrust. This results in a demonstrated throttling ratio of 6.82:1, a significant improvement over the 4:1 throttling ratio of the previous design.

The experimental steady state performance parameters are plotted versus chamber pressure and are compared with theoretical performance parameters in Figures 13 through 16. The experimental thrust versus chamber pressure curve (Figure 13) is about 3 pounds higher than the theoretical curve except at the lower chamber pressures where the two curves converge. It is suspected that the effect of the assumption of zero bulk velocity of the gases in the combustion chamber (pg. 13) causes the two curves to diverge at the higher chamber pressures, because 3 pounds of thrust are produced when the propellants are run through the engine at full throttle without ignition.

The experimental characteristic exhaust velocity (Figure 14) decreases from 92.6% of theoretical at 1/8 throttle to 86.3% of theoretical at full throttle. This result is probably caused by more complete combustion of the propellants at the lower throttle settings due to the longer stay times at the lower thrust levels. Stay time is defined as the average time spent by each gas molecule within the chamber volume (Ref. 8).

The experimental thrust coefficient is higher than the theoretical thrust coefficient over the entire throttling range (Figure 15). When this result was observed, error in the experimental data was suspected, so the chamber pressure and thrust instrumentation was recalibrated; however, no deviations from the previous calibration curves were detected. The data reduction computer program was also checked and found to be correct. Thus, there is either error in the calculation of the theoretical data or invalid assumptions were made. The comparison of experimental to theoretical specific impulse (Figure 16) is, of course, merely the combination of the two previous results discussed.

Transient Operation

The investigation of the transient response of the throttleable engine was concerned with determining how closely

the thrust variation followed the throttle position.

The thrust and throttle position measurements were obtained directly from electrical signals which are nearly instantaneous in contrast to the pneumatic signals involved in obtaining the chamber pressure and flow rate measurements. The thrust curves for the transient runs are plotted and compared to the steady state thrust profile in Figure 17 . No difference in the transient results could be detected between the one second or the two second transient runs. It was noted that the force of the hydraulic actuator on the throttle arm added to the thrust reading while the throttle was being opened, and that the thrust was reduced while the throttle was being closed. It was impossible to accurately measure this effect, so it is mentioned here qualitatively and not accounted for in the plotted values of transient thrust.

VI. Conclusions

1. The results of this investigation verify that the type of injector area variation used in this study is a satisfactory method of varying the thrust of a gaseous propellant rocket engine.

2. The thrust of the engine can be varied continuously from 11 to 75 pounds while its specific impulse remains nearly constant.

3. The engine demonstrates rapid and accurate response and gives no indication of combustion instability.

4. The water cooling provided sufficiently limits wall temperatures for runs up to at least 8 seconds duration.

5. The system provides a stable and reliable tool for further research on the throttling problem.

6. The mounting of the throttle actuator on the test stand prevents an accurate determination of the transient thrust curve.

VII. Recommendations

Operational throttleable rocket engines will probably employ regenerative cooling techniques. This may present severe heat transfer problems at low thrust levels when the propellant (which is also the coolant) mass flow rates greatly decrease and the flame temperature remains nearly constant. Assuming that the fuel is used as the coolant, it would be desirable to operate at lower mixture ratios at low thrust settings and higher mixture ratios at higher thrust settings. Therefore, it is suggested that studies be made to develop mixture ratio programs and that they be tested on the existing engine.

Attempts should also be made to improve the thrust profile of the engine to make the thrust linear with respect to the throttle position. Both of the above suggestions could be accomplished by redesign of the propellant orifice geometries.

The existing engine could also be used for a variety of nozzle design studies. If any such projects are attempted, it is suggested that a smaller throat diameter be used in order to conserve the test facility gas supply.

Finally, in the interest of obtaining more accurate experimental data, a method of controling the throttle position without influencing the thrust measurement should be designed. This could be done by mounting a hydraulic or electric actuator

on the engine itself as opposed to mounting it on the test stand as it was during this investigation.

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Table I

Steady State Runs at 12.5% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6815	34.5	2.49	131.4	6910.	1.27	271.7
7908	41.5	2.20	144.4	7889.	1.38	339.6
8009	36.5	2.57	130.4	7314.	1.35	306.8
8008	36.0	2.37	132.4	7540.	1.31	307.2
8117	34.0	2.23	128.4	7637.	1.28	302.9
8109	37.5	2.24	138.4	7536.	1.31	305.8
8721	34.5	2.17	130.4	7881.	1.28	312.3
8708	33.5	2.31	124.4	8266.	1.30	333.5
8817	34.0	2.43	128.4	8066.	1.28	319.8
8808	35.0	2.25	134.5	8270.	1.25	322.5
8908	36.5	2.16	136.4	8039.	1.29	322.2
Average	35.8	2.31	133.0	7759.	1.30	313.1

Table II

Steady State Runs at 25% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6814	54.0	2.22	196.4	7520.	1.32	309.7
7907	56.0	2.01	192.4	7745.	1.40	337.6
8010	52.5	2.11	180.4	7338.	1.40	320.0
8007	55.0	2.13	189.4	7411.	1.40	322.4
8116	55.0	2.18	190.4	7482.	1.39	323.7
8108	53.0	2.09	192.4	7587.	1.33	313.0
8720	51.0	2.07	182.4	7551.	1.35	316.2
8707	52.5	2.20	194.4	7899.	1.30	319.6
8816	52.5	2.25	190.4	7912.	1.33	326.7
8807	52.0	2.14	185.5	7763.	1.35	326.0
8907	53.0	2.06	191.4	7925.	1.33	328.7
Average	53.3	2.13	189.6	7648.	1.39	322.1

Table III

Steady State Runs at 37.5% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6813	64.5	2.13	224.4	7485.	1.39	322.2
7906	64.5	1.92	220.4	7802	1.31	342.0
8011	63.5	2.18	214.4	7527.	1.43	334.0
8006	64.0	2.03	214.4	7520.	1.44	336.3
8115	62.5	2.21	214.4	7354.	1.40	321.1
8107	61.0	2.10	217.4	7456.	1.35	313.3
8719	59.5	2.25	210.4	7602.	1.36	322.0
8706	60.0	2.27	208.4	7703.	1.39	332.2
8815	60.0	2.19	214.4	7796.	1.35	326.7
8806	60.5	2.10	212.5	7735.	1.37	329.9
8906	61.0	2.01	215.4	8013.	1.36	339.9
Average	61.9	2.13	215.1	7636.	1.39	329.0

Table IV

Steady State Runs at 50% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6812	69.0	2.10	236.4	7714.	1.41	337.2
7905	72.0	1.81	238.4	7727.	1.46	349.5
8012	70.0	1.97	232.4	7435.	1.45	335.5
8005	69.0	1.98	226.4	7415.	1.47	338.6
8114	68.5	2.10	234.4	7550.	1.41	330.5
8106	67.0	2.07	235.4	7498.	1.37	319.6
8718	67.5	2.08	227.4	7494.	1.43	333.2
8705	66.0	2.21	224.4	7375.	1.42	325.0
8814	66.5	2.20	228.4	7672.	1.40	334.5
8805	65.5	2.16	226.5	7596.	1.39	329.1
8905	66.5	2.07	231.4	7771.	1.38	334.5
Average	68.0	2.07	231.0	7568.	1.42	333.4

Table V

Steady State Runs at 62.5% Throttle

Run Number	F, lb_f	MR	P_c, psia	C^*, fps	C_f	I_{sp}, lb_f sec/lb_m
6811	71.0	2.09	246.4	7628.	1.39	329.2
7904	75.0	1.85	248.4	7670.	1.46	346.9
8013	73.0	2.12	240.4	7251.	1.46	329.9
8004	71.0	1.98	235.4	7327.	1.45	331.1
8113	70.5	2.00	244.4	7462.	1.39	322.4
8105	71.0	2.02	244.4	7525.	1.40	327.4
8717	66.5	2.16	231.4	7344.	1.39	316.1
8704	68.5	2.10	234.4	7519.	1.41	329.2
8813	70.0	2.18	234.4	7599.	1.44	339.9
8804	68.5	2.14	238.5	7694.	1.38	331.1
8904	68.5	2.10	240.4	7674.	1.37	327.5
Average	70.3	2.07	239.9	7518.	1.41	330.1

Table VI

Steady State Runs at 75% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6810	73.5	2.05	250.4	7445.	1.41	327.3
7903	76.0	1.79	252.4	7663.	1.45	345.6
8014	72.0	2.02	238.4	7297.	1.46	330.2
8003	72.0	1.81	240.4	7416.	1.45	335.1
8112	71.0	1.81	246.4	7585.	1.39	327.4
8104	72.5	1.98	250.4	7528.	1.40	326.5
8716	71.5	2.01	240.4	7418.	1.43	330.5
8703	69.0	1.89	234.4	7522.	1.42	331.7
8812	71.5	2.13	242.4	7649.	1.42	337.9
8803	69.5	1.90	238.5	7667.	1.40	334.7
8903	71.5	1.97	245.4	7665.	1.40	334.5
Average	71.8	1.94	243.6	7532.	1.42	332.8

Table VII

Steady State Runs at 87.5% Throttle

Run Number	F, lb_f	MR	P_c, psia	C^*, fps	C_f	I_{sp}, lb_f sec/lb_m
6809	74.5	2.19	253.4	7488.	1.42	329.7
7902	76.5	1.89	252.4	7748.	1.46	351.0
8015	74.5	2.03	246.4	7351.	1.46	333.0
8002	74.5	1.81	242.4	7279.	1.48	335.2
8111	74.0	1.84	252.4	7580.	1.41	332.6
6103	74.0	1.99	254.4	7555.	1.40	329.1
8715	73.0	2.01	244.4	7491.	1.44	335.2
8702	72.0	2.02	250.4	7790.	1.39	335.6
8811	71.5	2.00	242.4	7626.	1.42	336.9
8802	72.0	1.94	242.5	7606.	1.43	338.3
8902	73.0	1.89	250.4	7721.	1.40	337.1
Average	73.6	1.96	248.3	7567.	1.43	335.9

Table VIII

Steady State Runs at 100% Throttle

Run Number	F, lb_f	MR	P_c, psia	C*, fps	C_f	I_{sp}, lb_f sec/lb_m
6808	74.5	2.24	252.4	7233.	1.42	319.7
7901	79.0	1.85	256.4	7605.	1.48	351.0
8016	75.0	1.93	246.4	7248.	1.47	330.5
8001	76.0	1.93	246.4	7194.	1.49	332.4
8110	74.0	1.85	251.4	7364.	1.42	324.6
8101	77.5	1.93	256.5	7517.	1.46	340.2
8714	74.5	2.07	248.4	7339.	1.45	329.7
8701	72.5	1.98	250.4	7484.	1.40	324.6
8810	73.0	2.19	244.4	7441.	1.44	332.9
8801	72.5	1.97	244.5	7405.	1.43	329.0
8901	74.0	1.98	251.4	7476.	1.42	329.6
Average	74.8	1.99	249.9	7391.	1.44	331.3

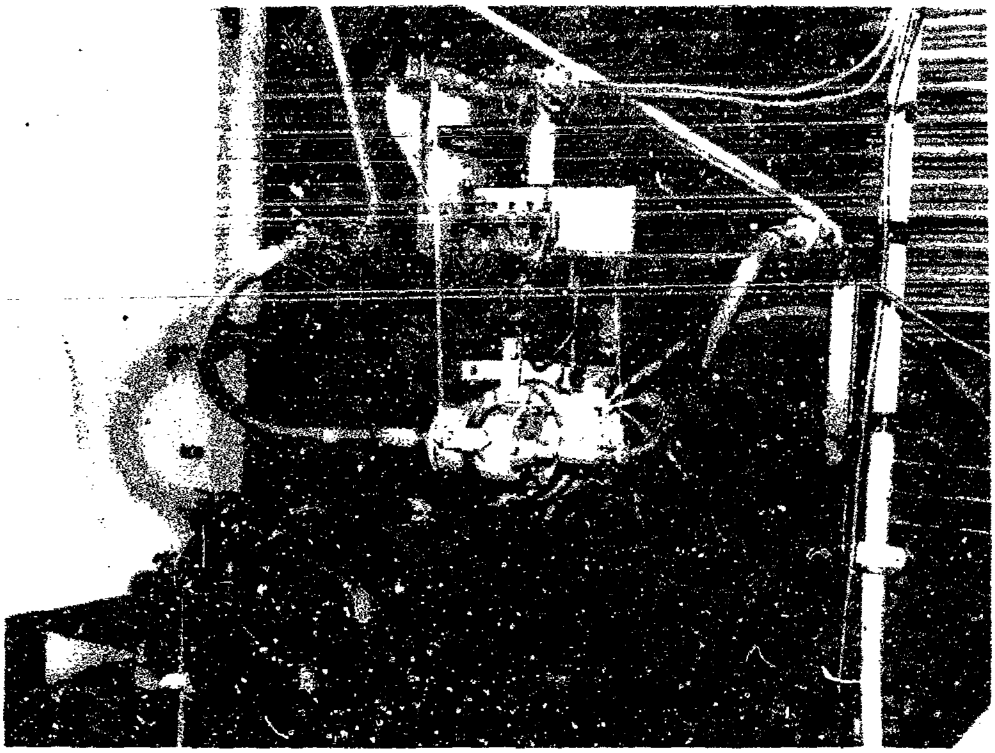

Figure 1

Rocket Engine Assembled on Test Stand

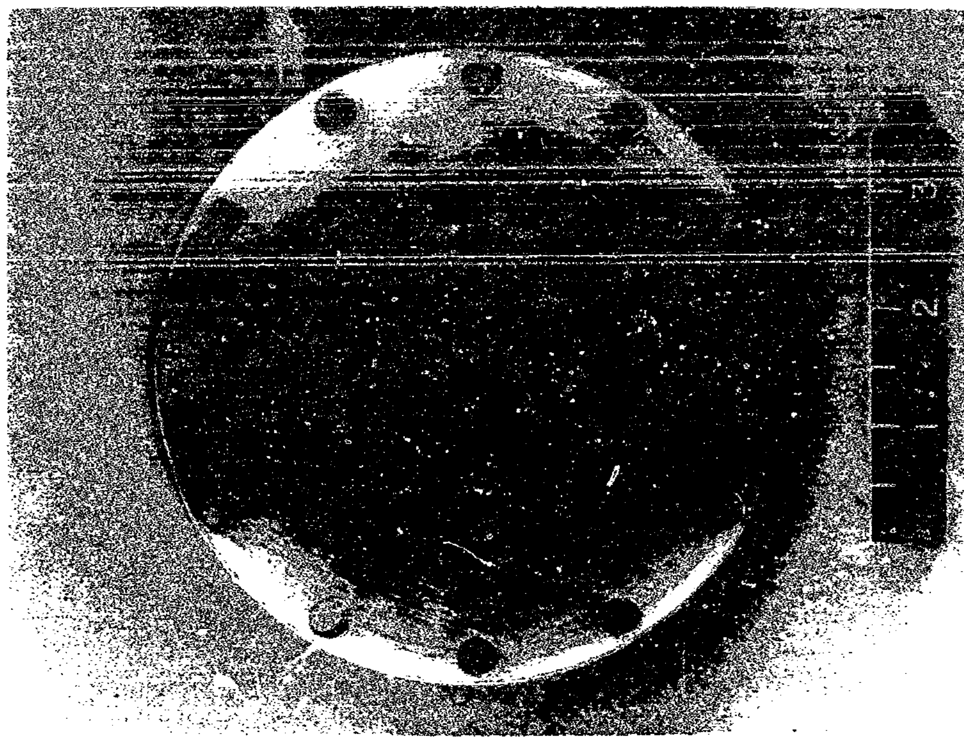

Figure 2

Injector Face

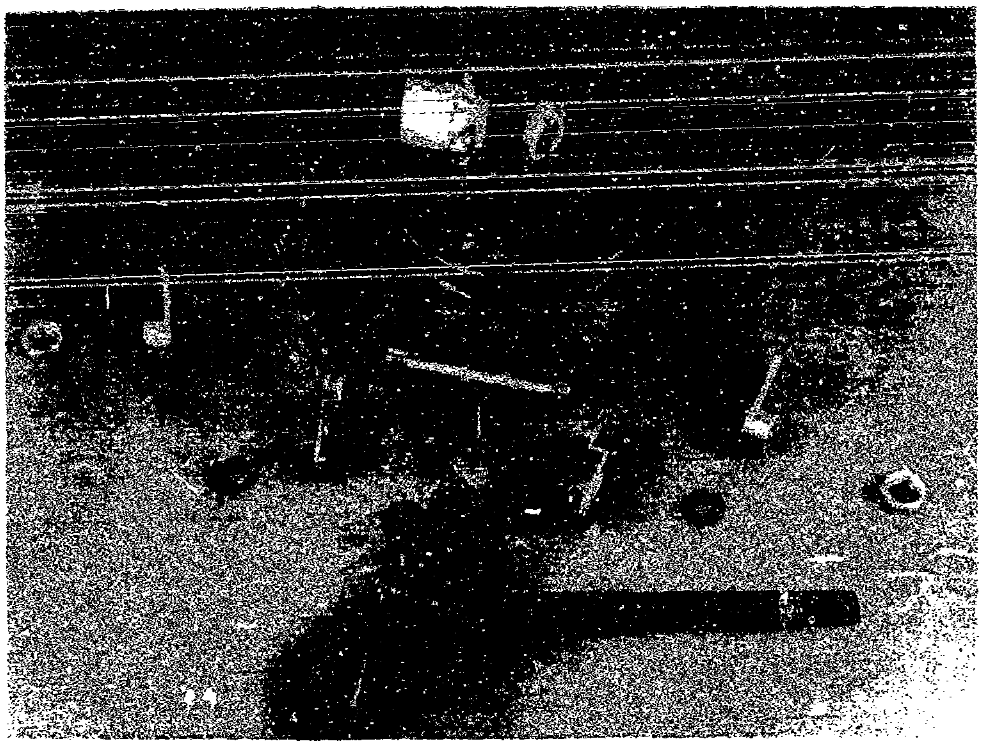

Figure 3

Injector Assembly

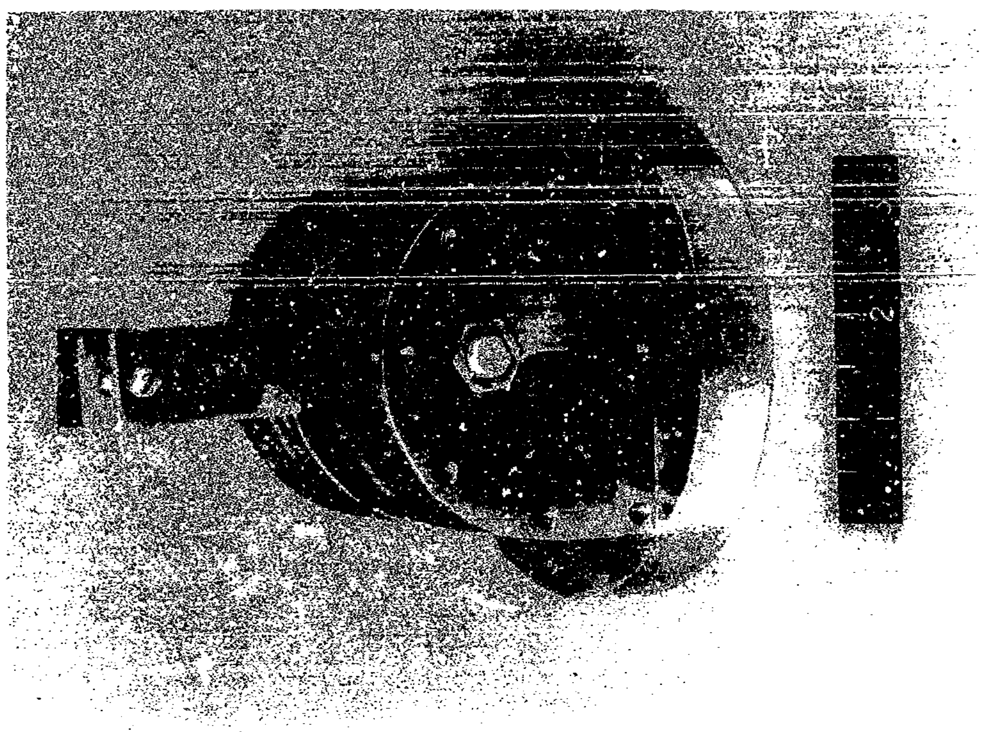

Figure 4

Assembled Injector

Figure 5

Injector, Chamber, and Nozzle

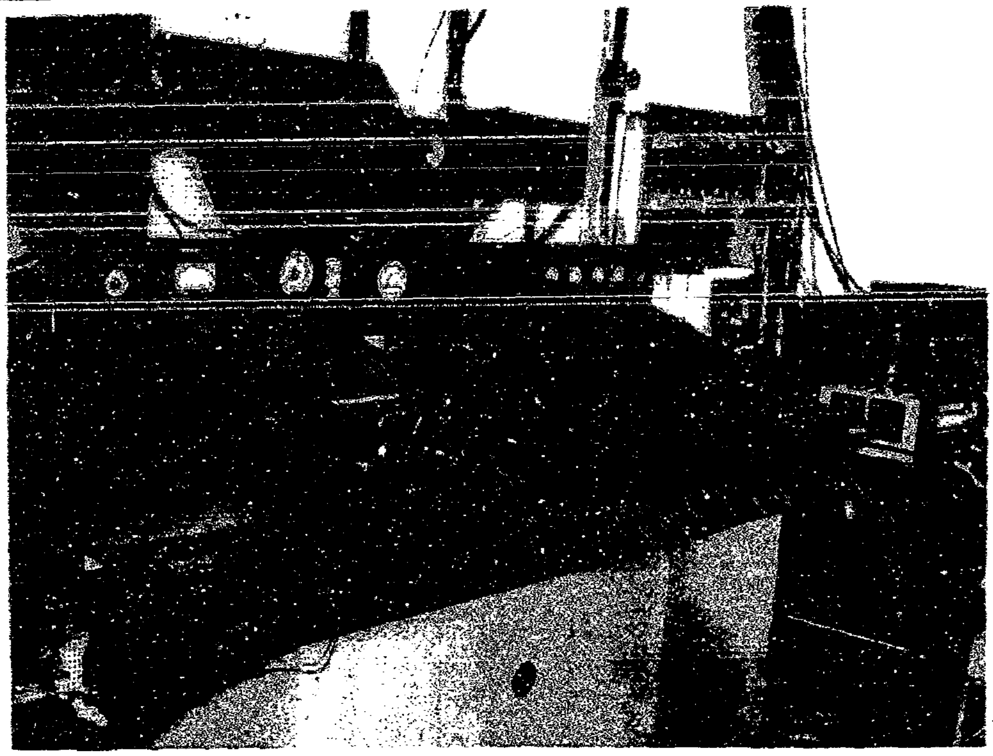

Figure 6

AFIT Rocket Test Facility Control Room

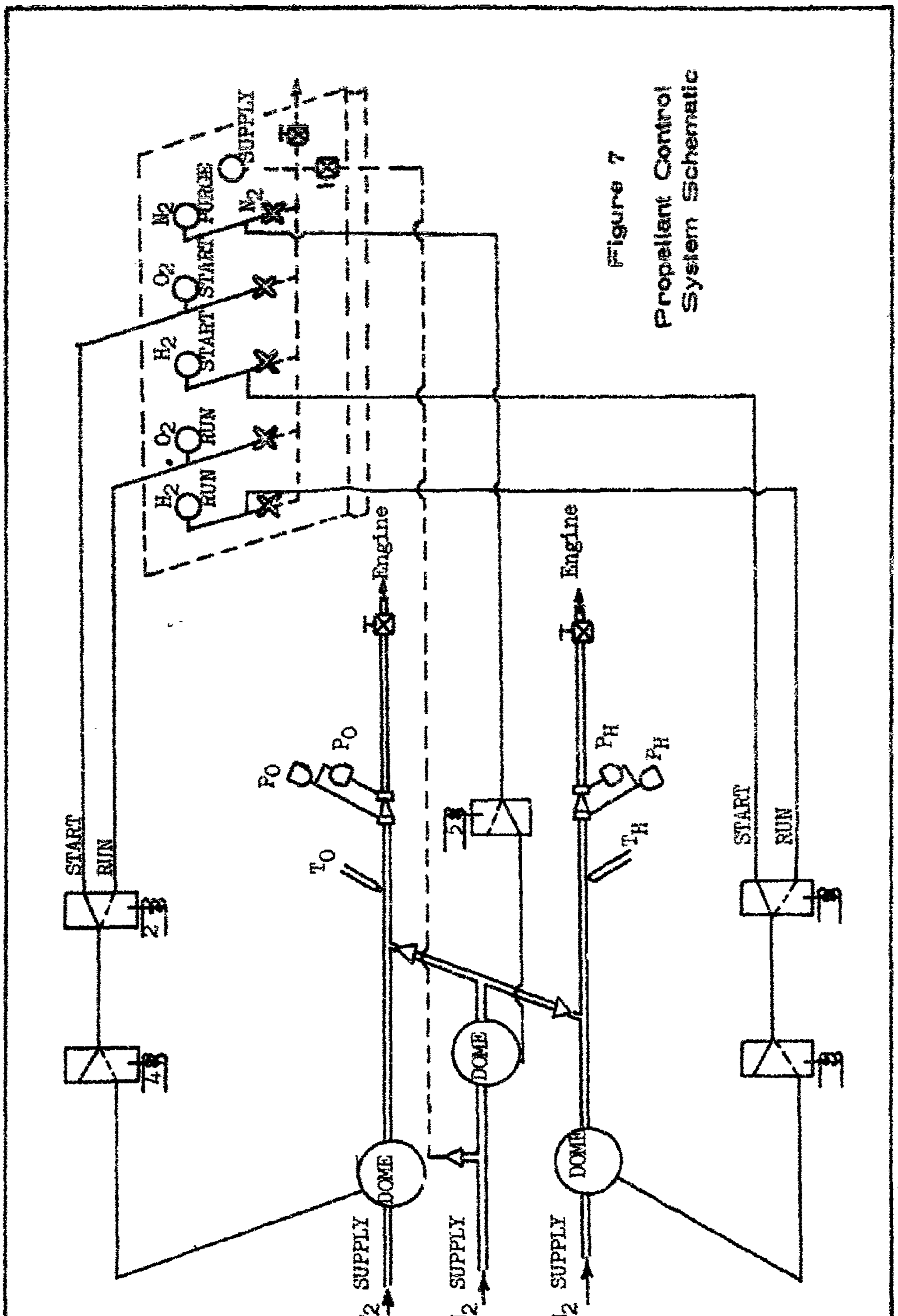

Figure 7
Propellant Control
System Schematic

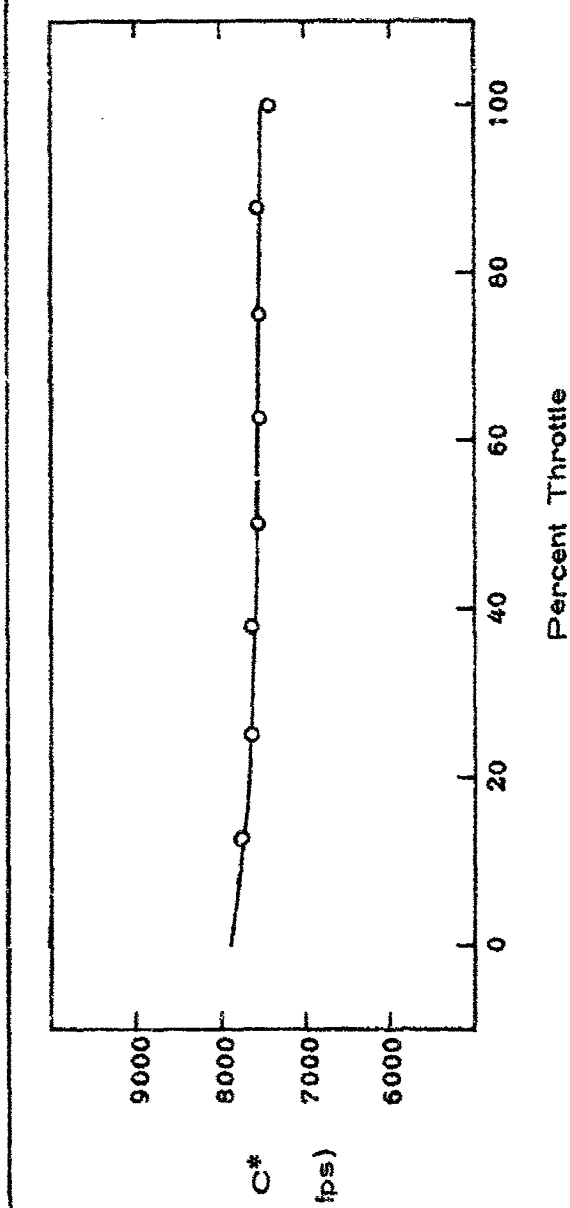

Figure 8

CHARACTERISTIC EXHAUST VELOCITY PROFILE

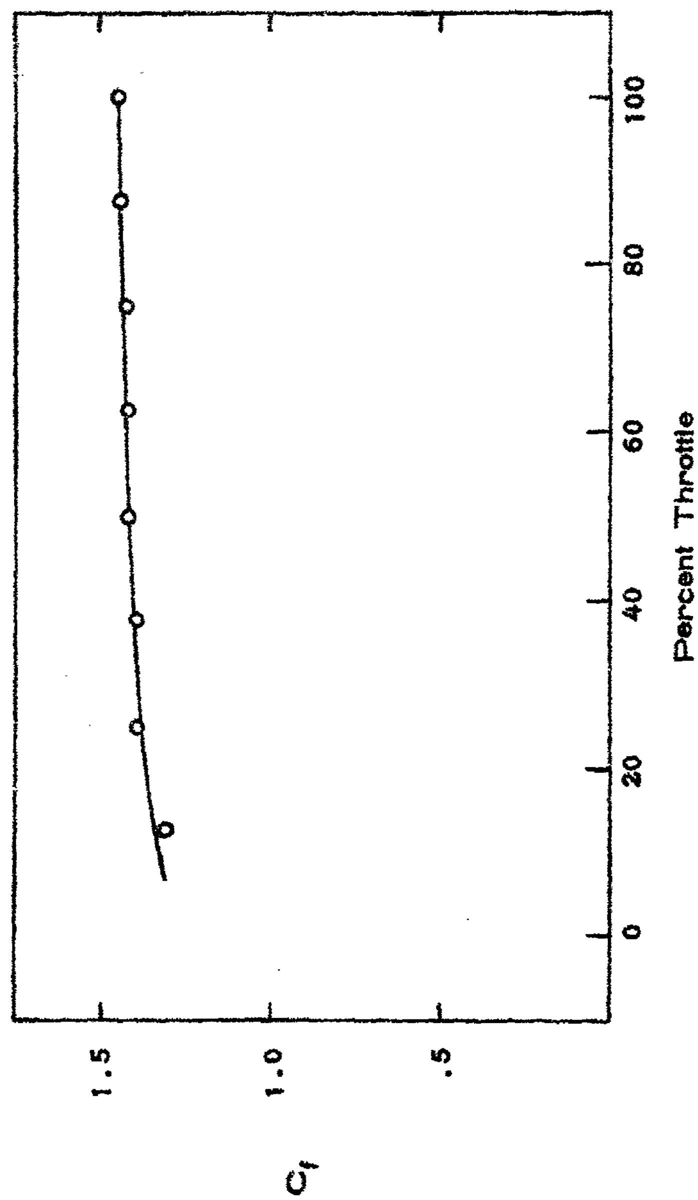

Figure 9

THRUST COEFFICIENT PROFILE

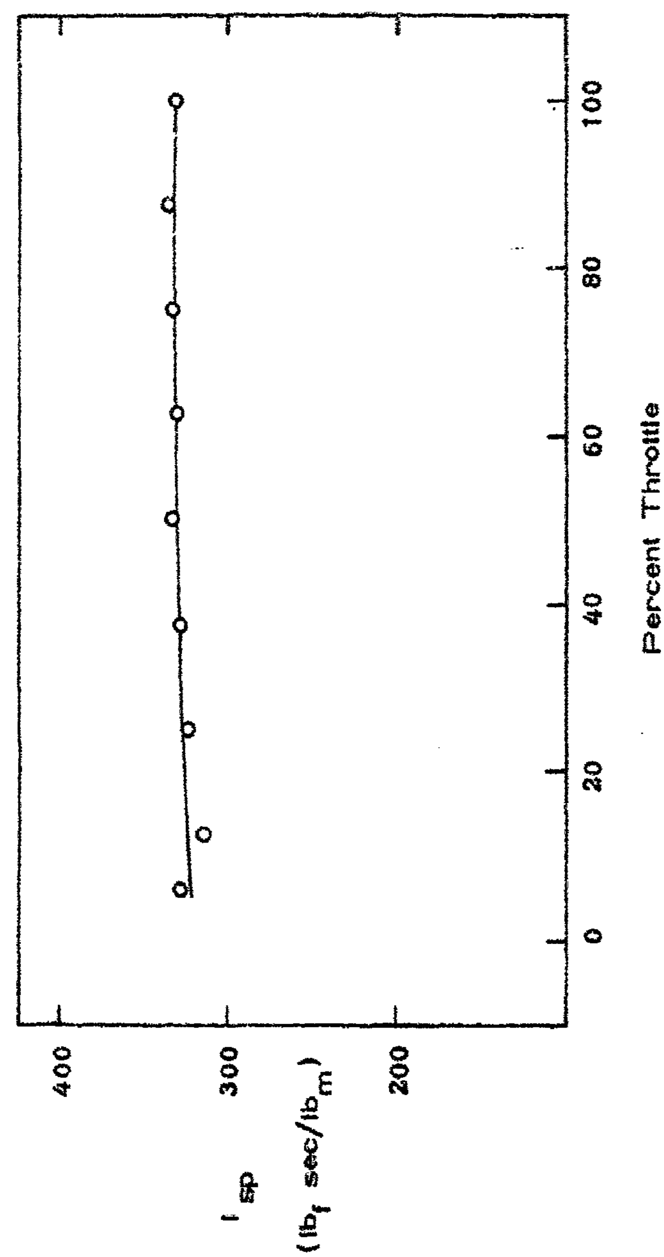

Figure 10

SPECIFIC IMPULSE PROFILE

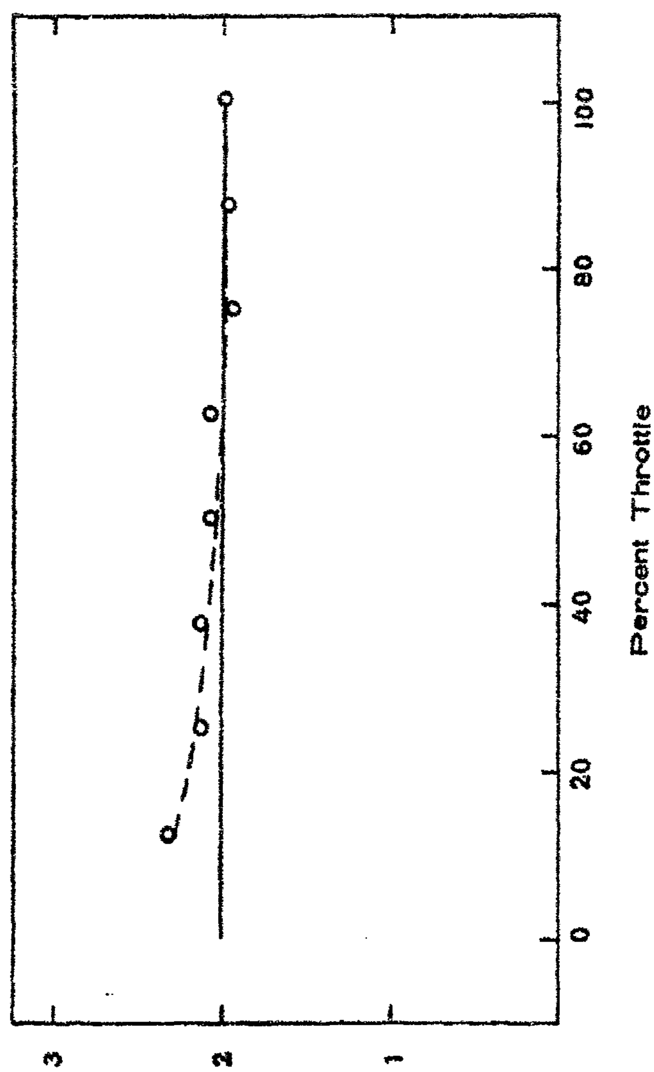

Figure 11

MIXTURE RATIO PROFILE

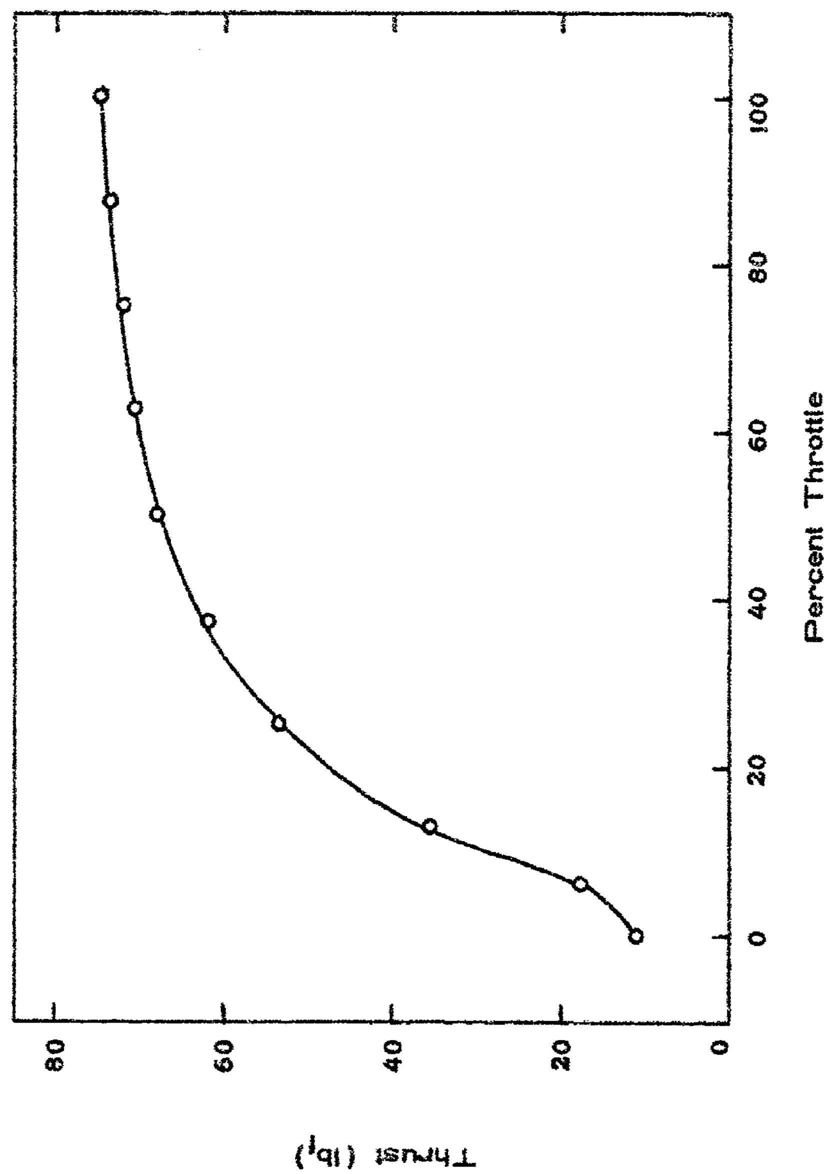

Figure 12
THRUST PROFILE

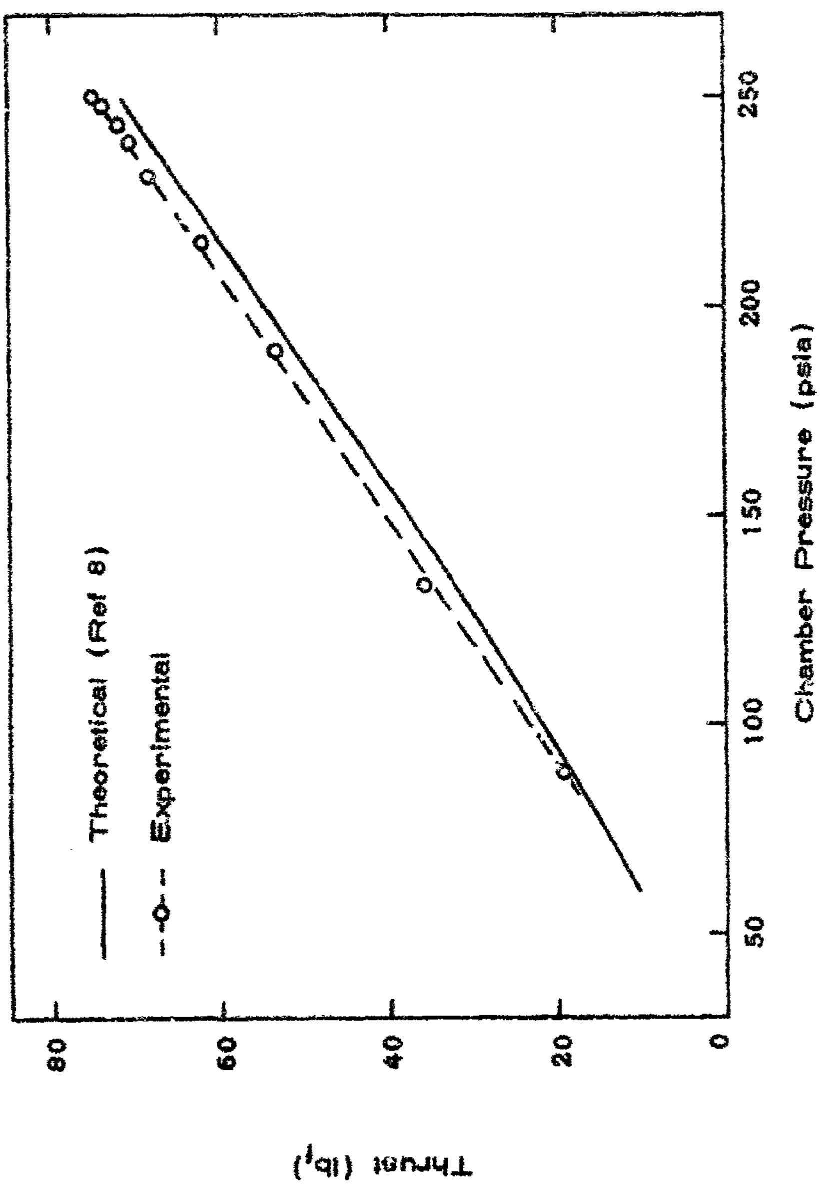

Figure 13

THRUST VS. CHAMBER PRESSURE

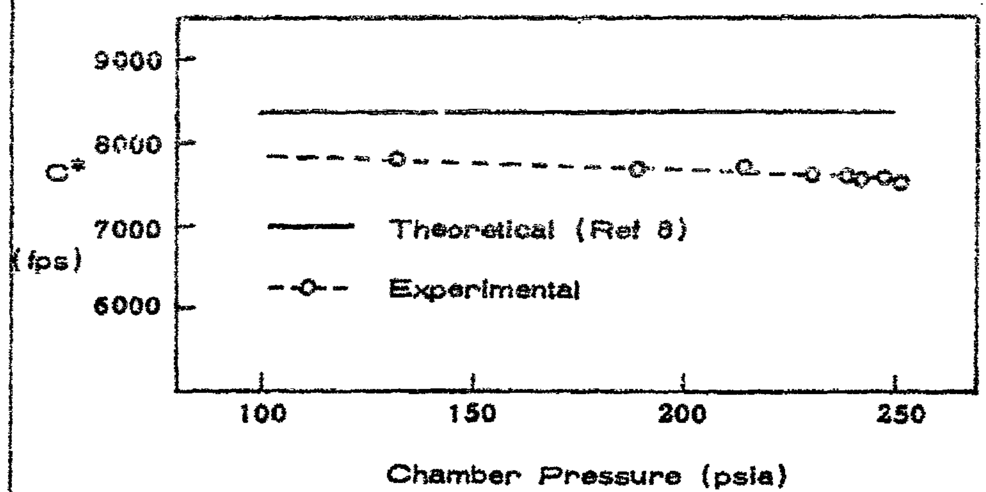

Figure 14

CHARACTERISTIC EXHAUST VELOCITY

VS. CHAMBER PRESSURE

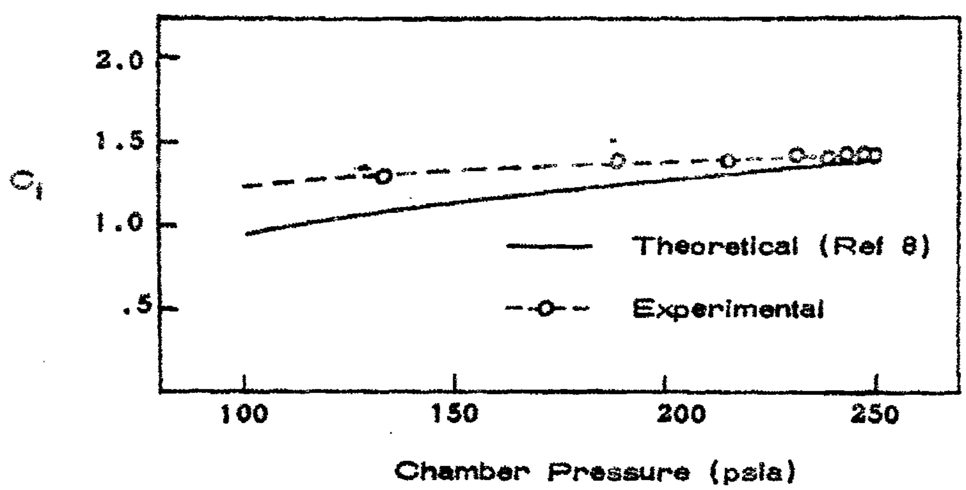

Figure 15

THRUST COEFFICIENT VS.

CHAMBER PRESSURE

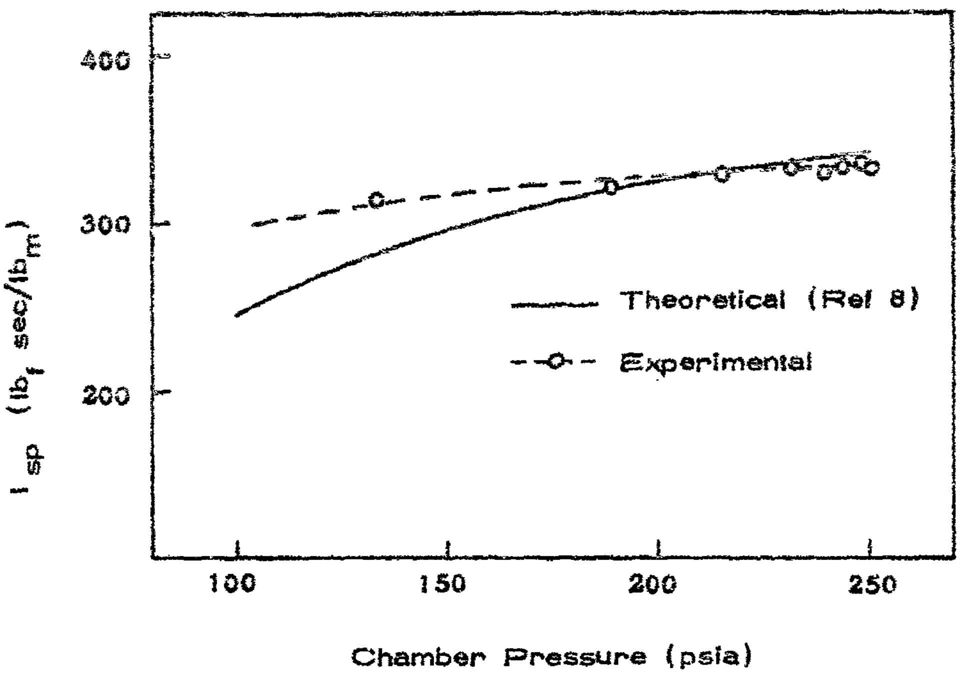

Figure 16

SPECIFIC IMPULSE VS. CHAMBER PRESSURE

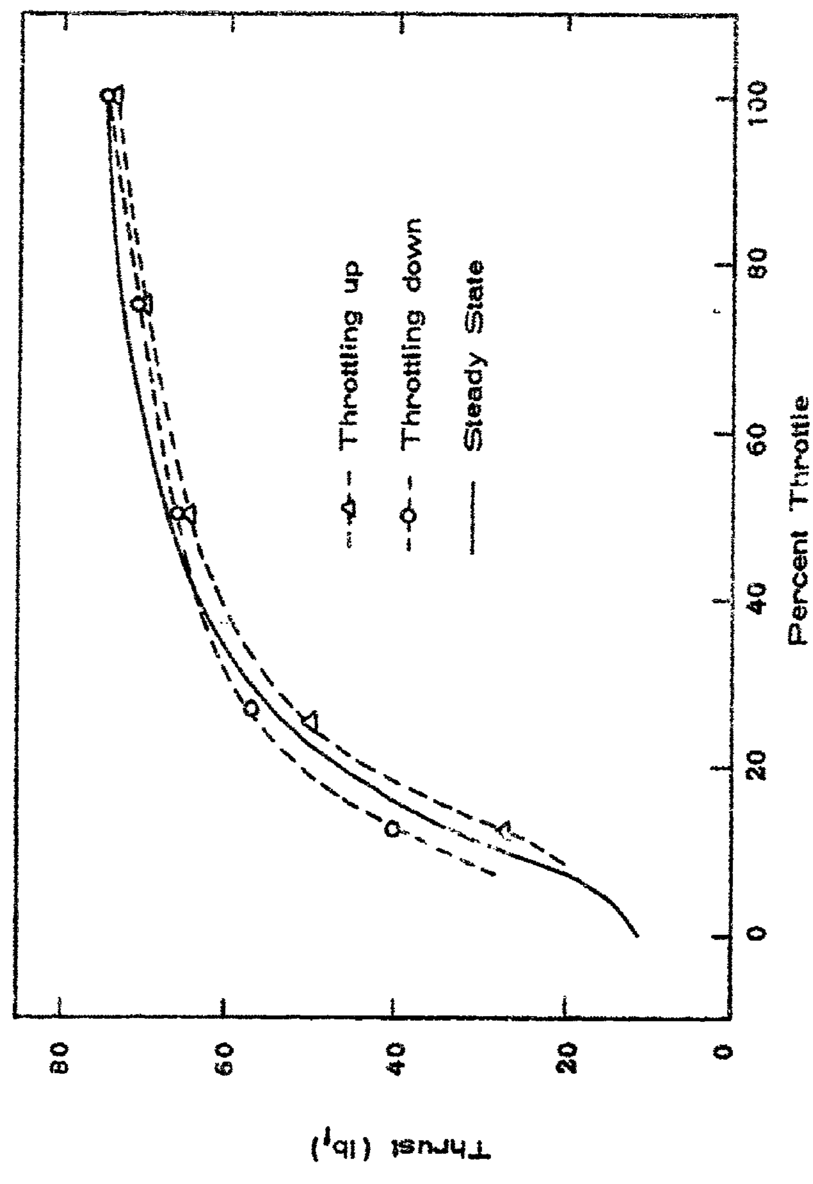

Figure 17

TRANSIENT THRUST RESPONSE

Appendix A

Injector Drawings

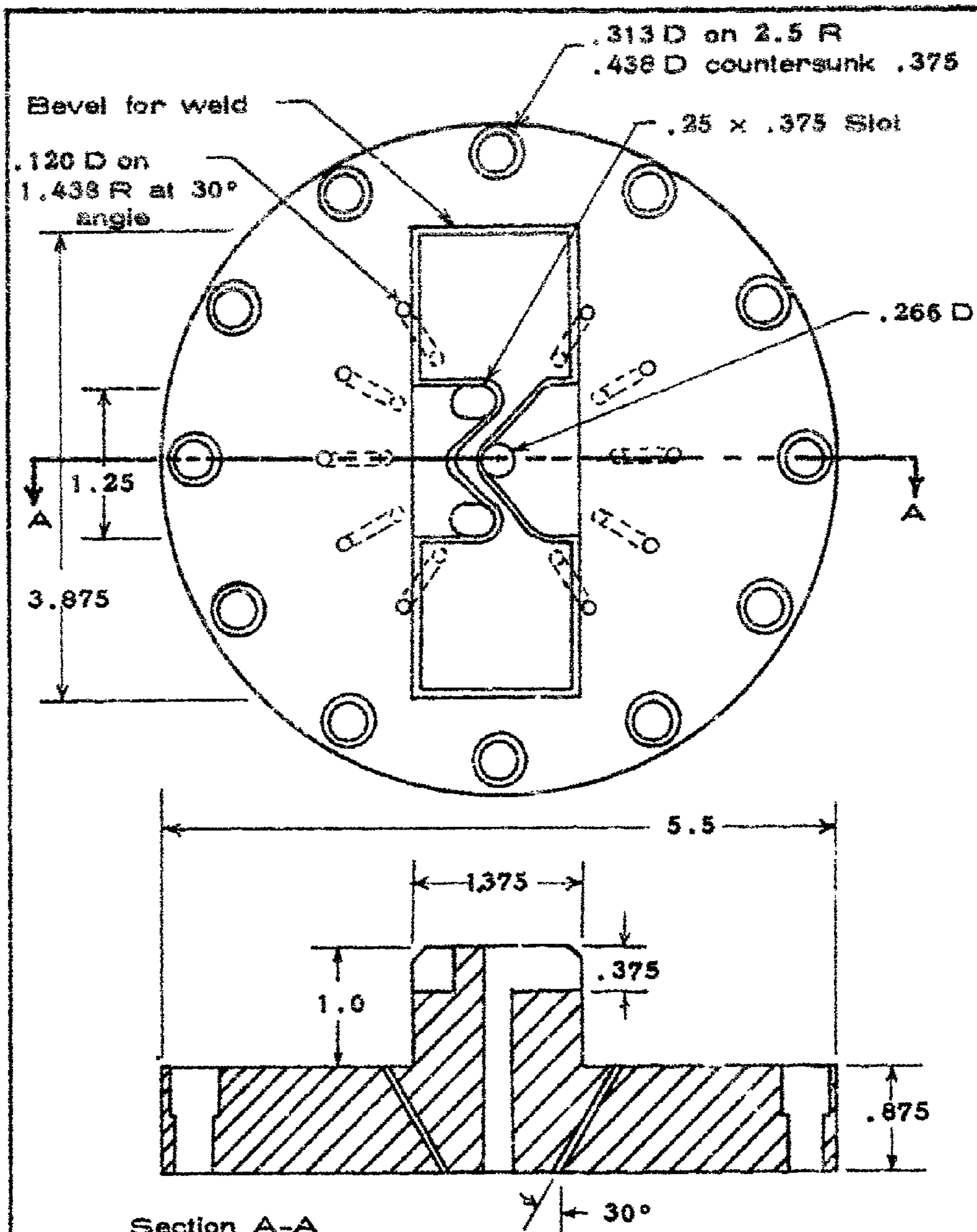

Figure 18

BASE PLATE

Scale: 3:4

1 Req'd

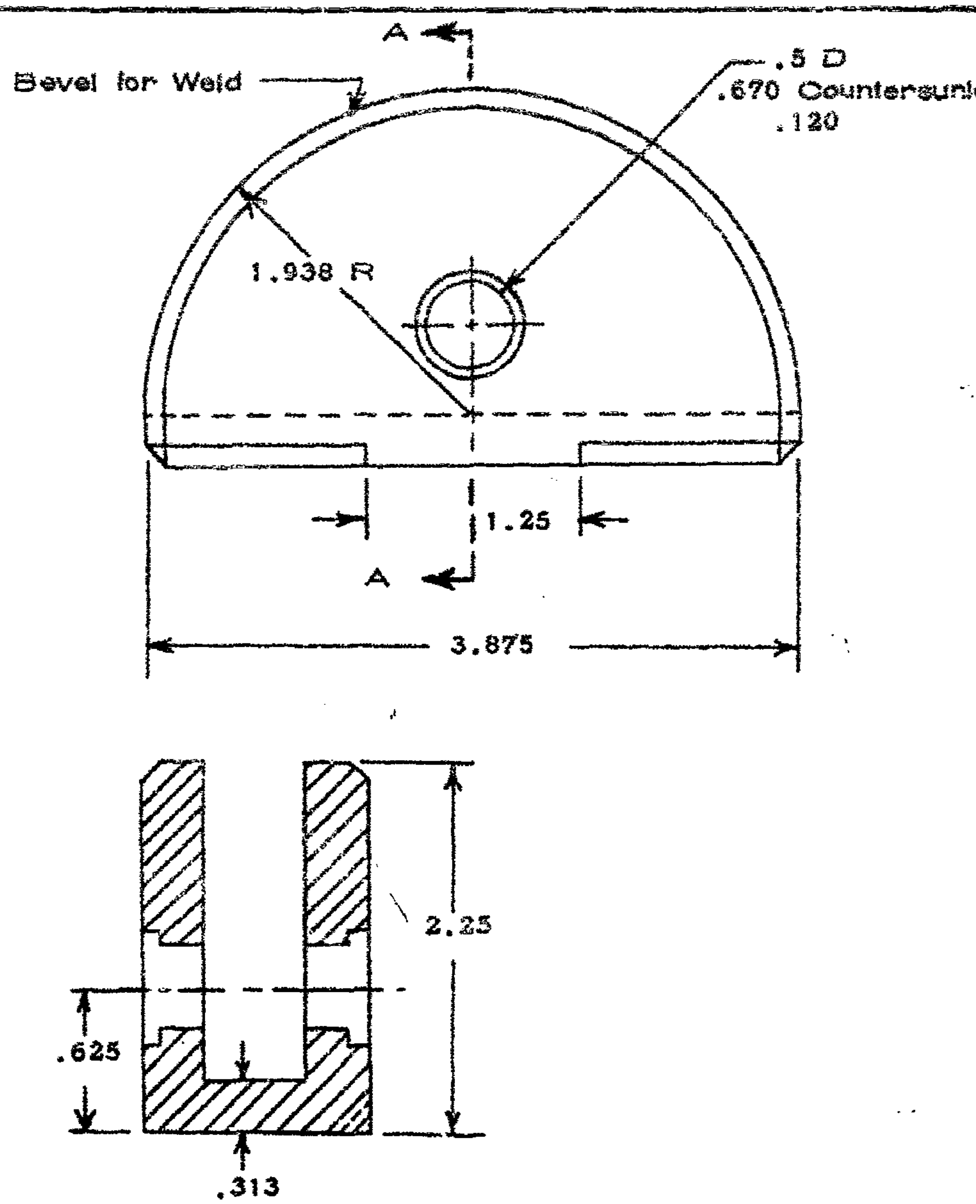

Figure 19

MANIFOLD SPACER

Scale: 1:1

1 Req'd

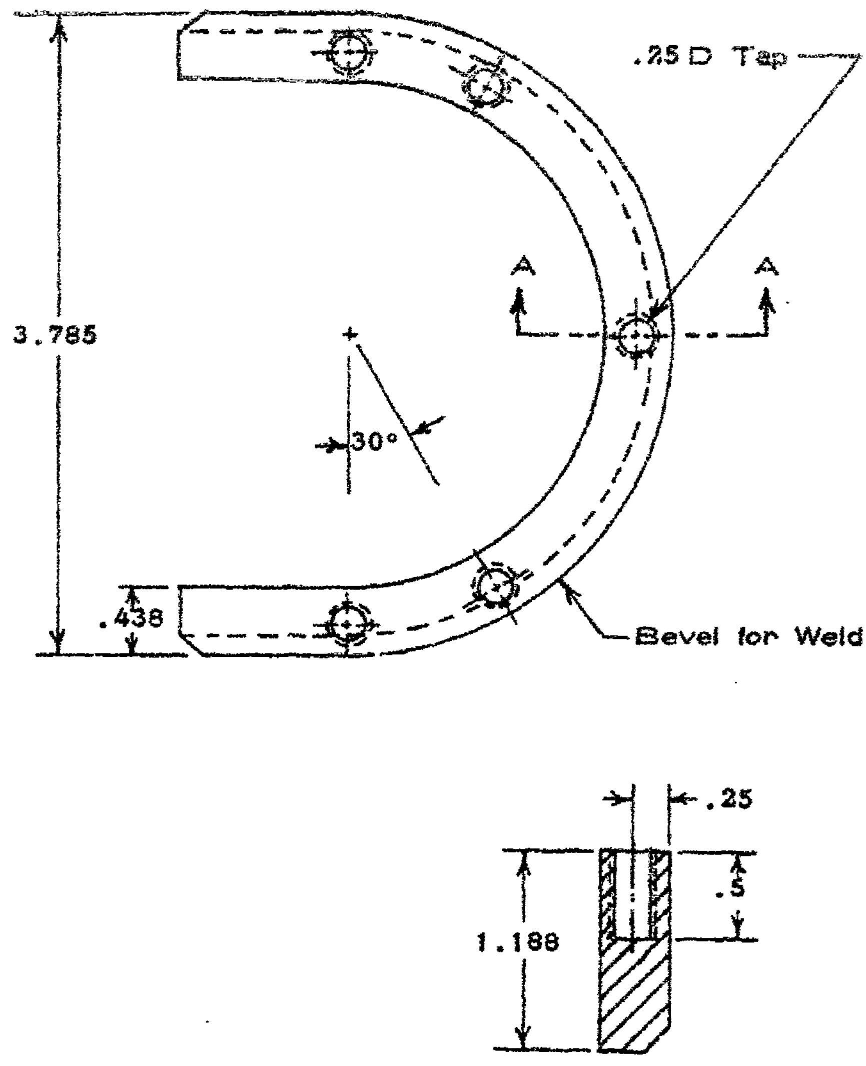

Figure 20
MANIFOLD JACKET
Scale: 1:1
2 Req'd

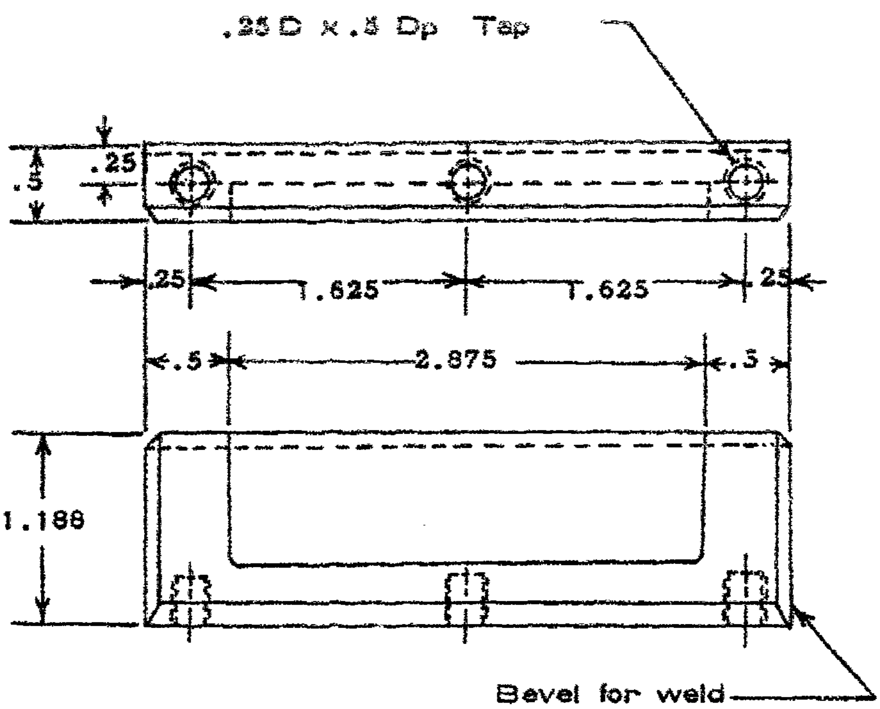

Figure 21

INJECTOR COVER

Scale: 1:1

2 Req'd

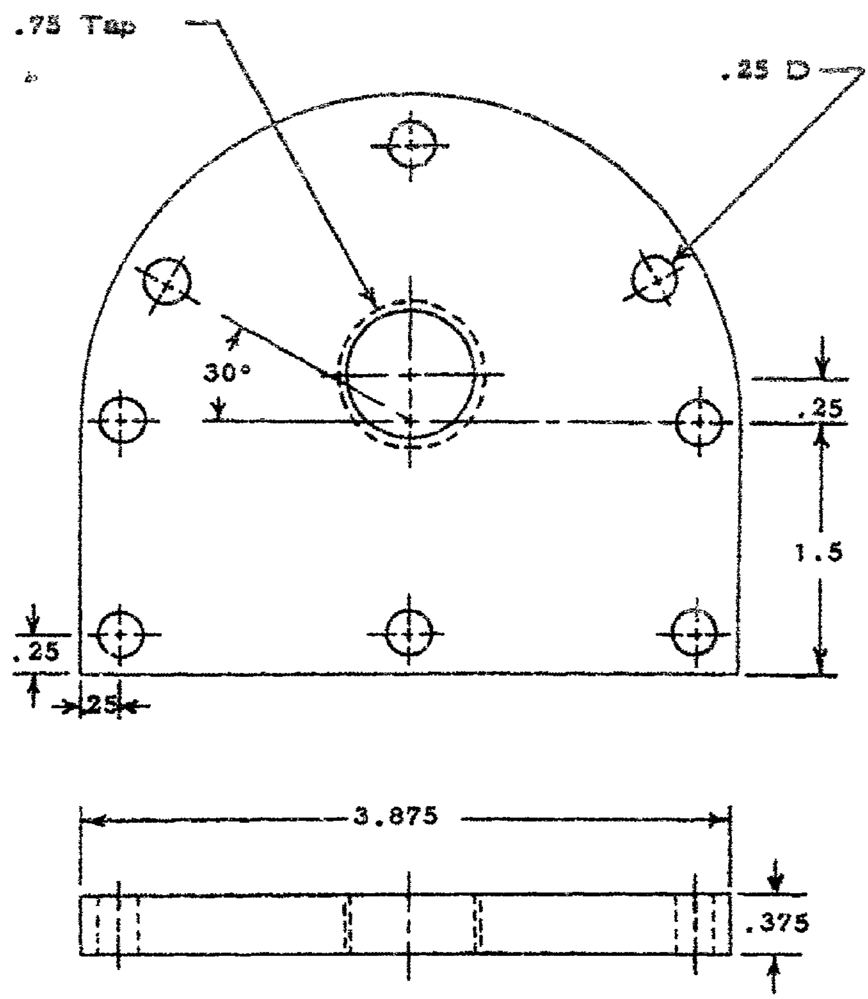

Figure 22

MANIFOLD COVER

Scale: 1:1

2 Req'd

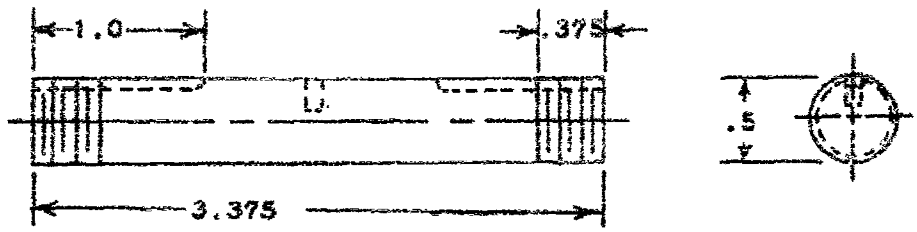

Figure 23

THROTTLE SHAFT

Scale: 1:1

1 Req'd

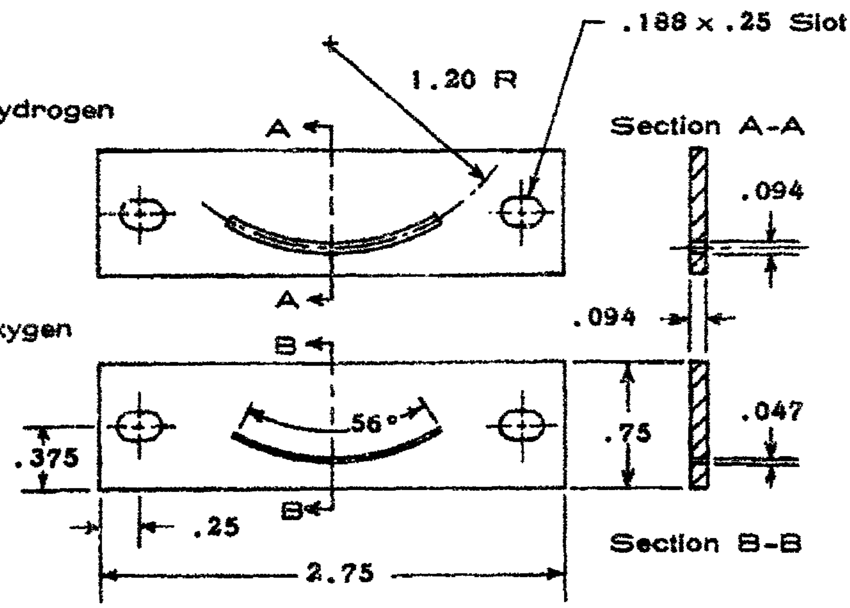

Figure 24

ORIFICE PLATES

Scale: 1:1

1 each Req'd

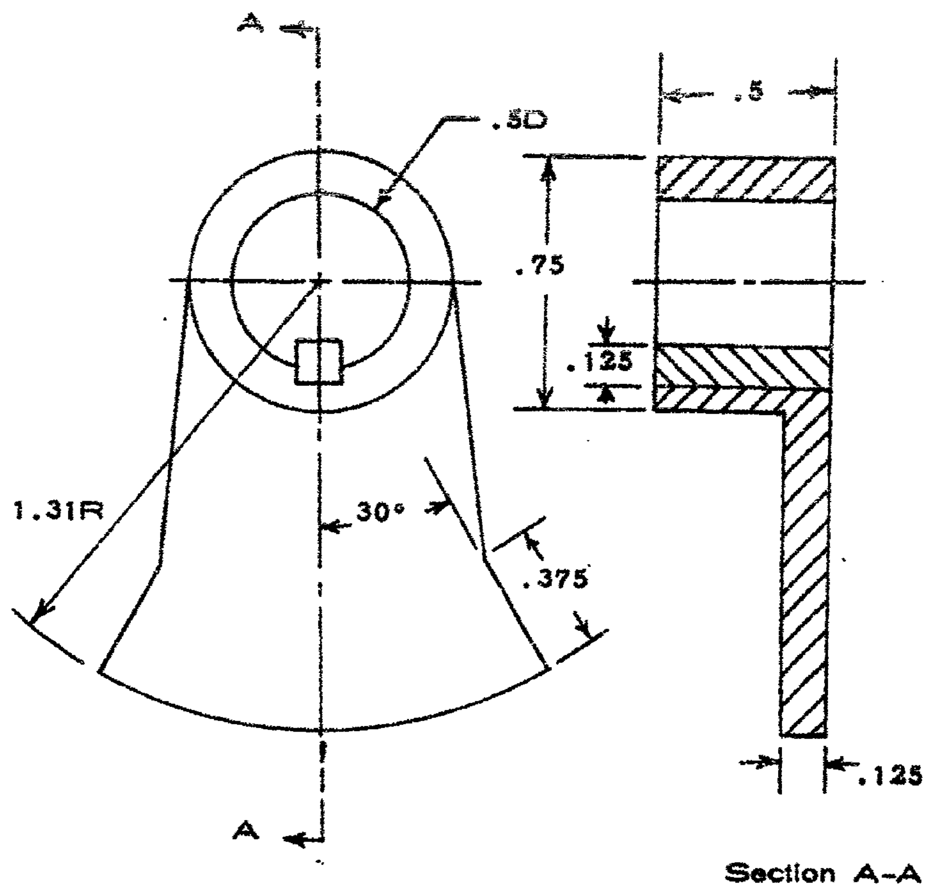

Figure 25

MOVEABLE PLATE

Scale: 2:1

2 Req'd

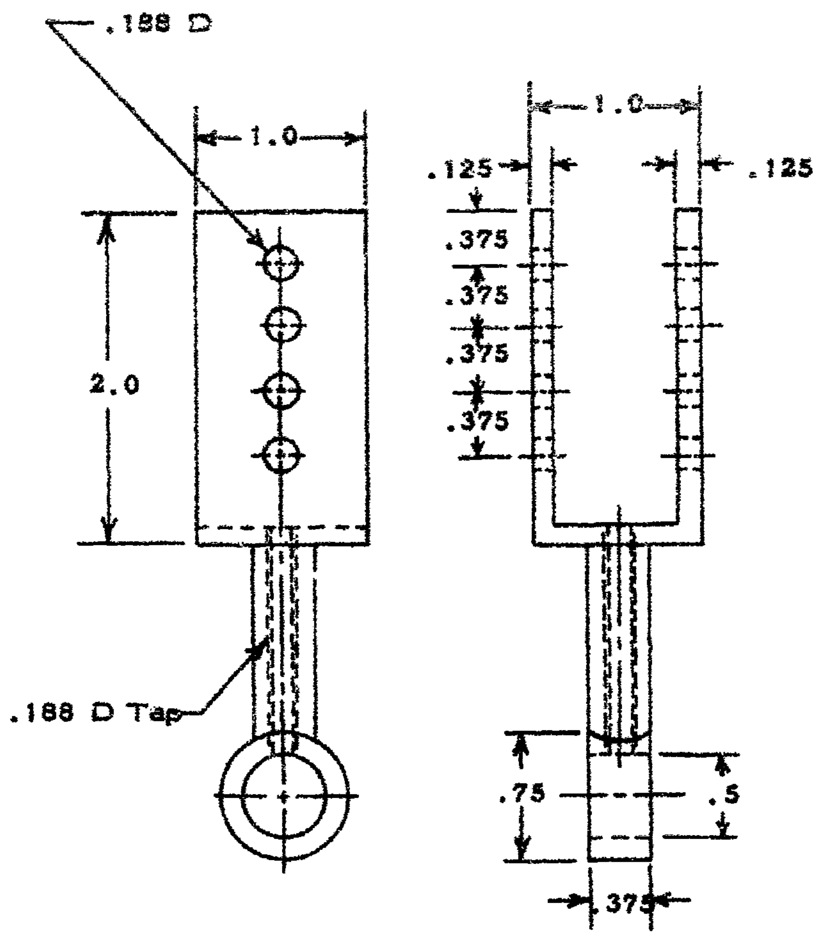

Figure 26

THROTTLE ARM

Scale: 1:1

1 Req'd

Appendix B

Hydraulic Control Mechanism

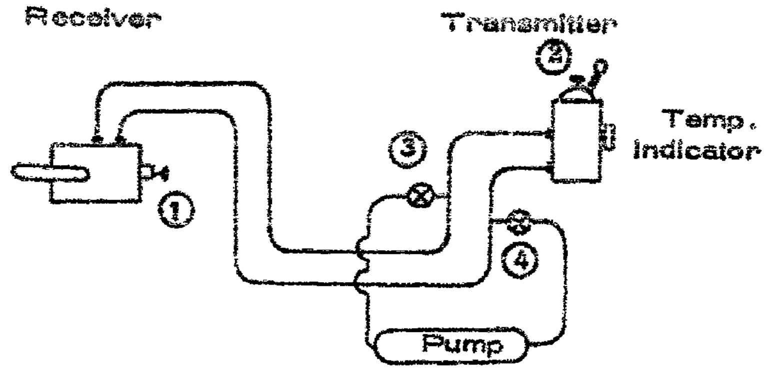

Figure 27

Hydraulic Control Schematic

Hydraulic Control System

The hydraulic exactor control system consists of a transmitter and a receiver connected by two 1/4 inch copper tubes as shown schematically above. This system provided positive manual control of the throttling mechanism, and proved to be well suited for its application in this investigation. A brief outline of the procedure used to prepare the system for operation follows, as an aid to anyone choosing to use the apparatus in future investigations.

To fill the lines and eliminate all air from the system, a hand operated hydraulic pump was used. Valves 1, 3 and 4 were opened and fluid was pumped through the lines until no evidence of air was present in the discharged fluid. Valves 1

and 2 are hex socket screws located as shown in figure 27.

Valve 1 was then closed, valve 2 opened, and fluid was again pumped through the lines to eliminate air. Then valve 3 was closed and the system was pressurized to approximately 50 psig, or until the temperature compensating indicator on the transmitter indicated approximately the ambient temperature of the system. Finally, valves 2 and 4 were closed and the pump was removed.

Appendix C

Data Reduction Program

```
C  C  DATA REDUCTION PROGRAM  DE GROOT
 701  FORMAT(I5,F6.1,F5.1,F6.4,F5.2,F6.1,F6.0,F5.2,F6.1)
 100  READ,N,PAHG,F,PCG,PHG,DPH,POG,DPO,TH,TO,PT
      AM=PAHG*.491
      PC=PCG+AM
      PH=PHG+AM
      PO=POG+AM
      RAD1=PH*DPH*560./TH
      FMH=.001419*(1.-.644*DPH/PH)*SQRT(RAD1)
      RAD2=PO*DPO*560./TO
      FMO=.001419*3.98*(1.-.644*DPO/PO)*SQRT(RAD2)
      FMT=FMO+FMH
      FMR=FMO/FMH
      AT=3.14159*.514*.514/4.
      CSTRX=PC*AT*32.174/FMT
      CFX=F/(PC*AT)
      SIMPX=F/FMT
      PUNCH701,N,PT,F,FMH,FMO,FMR,PC,CSTRX,CFX,SIMPX
      GO TO 100
      END
```

VITA

Frederick James De Groot was born on 19 January 1943 in Marinette, Wisconsin, the son of Clarence Albert De Groot and Phyllis Marie De Groot. After graduation from Port Washington High School, Port Washington, Wisconsin, he was appointed to the United States Air Force Academy, Colorado in June 1961. In June 1965 he graduated with the degree of Bachelor of Science and a commission as Second Lieutenant in the United States Air Force. He was assigned to the Air Force Institute of Technology as a full time graduate student in August 1965.

Permanent address: Ulao Parkway
Grafton, Wisconsin

This thesis was typed by Mrs. Jeannine De Groot.

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Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

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3. REPORT TITLE

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4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Thesis

5. AUTHOR(S) (First name, middle initial, last name)

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
None	Air Force Institute of Technology Air University

13. ABSTRACT

The purpose of this thesis was to investigate the performance of a variable thrust rocket engine incorporating a variable area injector. The injector was designed, constructed, and assembled on an existing thrust chamber which was lengthened and provided with water cooling. Gaseous hydrogen and oxygen were used as propellants. The thrust was varied over a continuous range from 11 to 75 pounds, resulting in a throttling ratio of 6.82:1, while the specific impulse remained nearly constant. The transient response of the engine was fast, smooth, and accurate, and no indication of combustion instability was observed during the investigation.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
THROTTLEABLE ROCKET						
GASEOUS PROPELLANT ROCKET						
THRUST VARIATION						
VARIABLE AREA INJECTOR						

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